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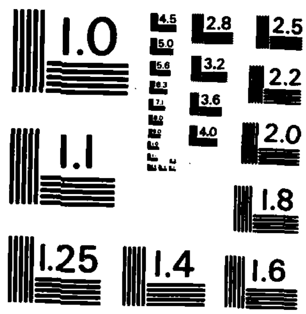
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computational fluid dynamics	vortex flows
compressor stall	inlet distortion

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A multi-investigator program on problems of current interest in turbomachinery fluid dynamics is being conducted at the M.I.T. Gas Turbine & Plasma Dynamics Lab. Within the scope of this effort, four different tasks, encompassing both design and off-design problems, have been identified. These are: 1) Investigation of fan and compressor design point fluid dynamics (including formation of design procedures using current three-dimensional transonic codes and development of advanced measurement techniques for use in transonic fans) 2) Studies of basic mechanisms of compressor stability enhancement using

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compressor casing/hub treatment; 3) Fluid mechanics of inlet vortex flow distortions in gas turbine engines; 4) Investigations of three-dimensional flows in highly loaded turbomachines (including blade-to-blade flow analysis and numerical computations of secondary flow in a bend using spectral methods. In additional to these tasks this multi-investigator effort also includes the Air Force Research in Aero Propulsion Technology (AFRAPT) Program.

This document describes work carried out on this contract during the period 10/1/81 - 4/30/82.

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CURRENT PROBLEMS IN TURBOMACHINERY FLUID DYNAMICS

for the period

October 1, 1981 to April 30, 1982

submitted to

AIR FORCE OFFICE OF SCIENTIFIC RESEARCH

Attention of:	Dr. James D. Wilson, Program Manager, Directorate of Aerospace Sciences, AFOSR (AFSC) Bolling Air Force Base, DC 20332
Principal Investigators:	Edward M. Greitzer William T. Thompkins, Jr. James E. McCune
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Collaborating Investigators:	Eugene E. Covert Sir William R. Hawthorne Frank H. Durgin

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1. INTRODUCTION AND RESEARCH OBJECTIVES

This report describes work carried out at the Gas Turbine and Plasma Dynamics Laboratory at M.I.T., as part of a multi-investigator effort on current problems in turbomachinery fluid dynamics. Support for this program is provided by the Air Force Office of Scientific Research under Contract Number F49620-82-K-0002, Dr. J.D. Wilson, Program Manager.

The present report gives a summary of the work for the period 10/1/81 - 4/30/82. For further details and background, the referenced publications and the Final Report (covering the period 10/1/79 - 9/30/81)[1] should be consulted.

Within the general topic, four separate tasks are specified. These have been described in detail in Reference 1, but they are, in brief;

1. Investigation of fan and compressor design point fluid dynamics
2. Studies of compressor stability enhancement
3. Fluid mechanics of gas turbine engine operation in inlet flow distortion
4. Investigations of three-dimensional flows in highly loaded turbomachines

In addition to these tasks, this year's effort also encompasses the Air Force Research in Aero Propulsion Technology program. The work carried out in each of the tasks will be described in the next section.

2. WORK TO DATE AND STATUS OF THE RESEARCH EFFORT

Task I: Investigations of Fan and Compressor Design Point Fluid Dynamics

A: Inverse Design Calculation

The first phase of the inverse design project emphasized development of at least a plausible inverse scheme for mixed supersonic, subsonic flow with the possibility of shock waves appearing in the flow field. In addition, even though this first phase studied only two-dimensional flow, no assumptions or techniques limiting the range of application to two-dimensional flows were to be made. This phase has now largely been completed with the publication of two technical papers: 1) "Inverse or Design Calculations for a Nonpotential Flow in Turbomachinery Blade Passages," in Journal of Eng. for Power, April 1982, and 2) "A Design Calculation Procedure for Shock-Free or Strong Passage Shock Turbomachinery Cascades," ASME paper 82-GT-220.

The next phase of this project was expected to focus on the numerical optimization itself, but preliminary investigation of known optimization techniques showed that our numerical solvers were unable to evaluate objective functions such as total pressure loss or mass flow rate accurately enough. As a result most of our work since the last reporting period has concentrated on improving the accuracy of our numerical flow field solvers. The fundamental problem was found to be the computation and interpretation coordinate transform metric terms appearing in the strong conservation law of the Euler equations. A proper interpretation of these terms allows a unification of the finite difference, strong conservation law approach and the finite volume approach. A proper computation of these terms allows much more accurate steady state solutions to be computed as illustrated in Figures 1.1 and 1.2. These figures show two computations for the same precompression type geometry.

In the first cáse, an incorrect oblique, trailing edge shock exists, while, in the second case, this shock is accurately resolved as a normal shock.

These algorithm refinements now allow us to compute, for the first time, accurate solutions for transonic supercritical compressor cascades and turbines cascades with the same computer codes used for supersonic compressor cascade calculations. Figure 1.3 shows a comparison between computed and experimental measurements for a turbine cascade. Quite satisfactory agreement exists over the entire surface, but the agreement is particularly good for the blunt leading edge stagnation point region which is traditionally most difficult to achieve.

It is expected that only a modest amount of further algorithm development is required before efforts can be shifted back to the numerical optimization process.

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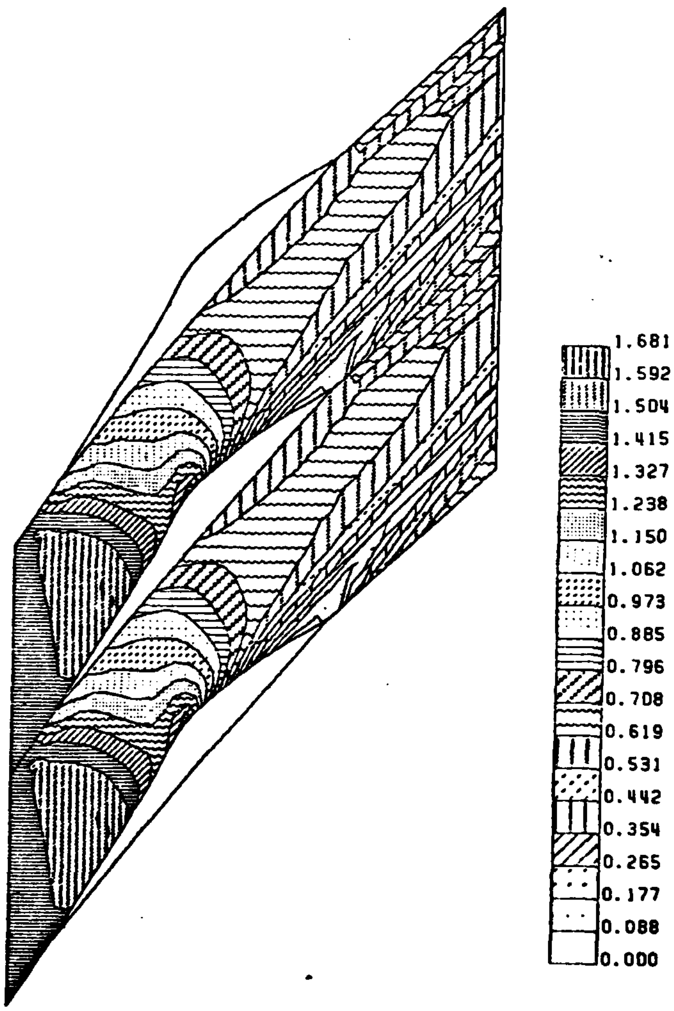

Figure 1.1 **PRE-COMPRESSION BLADE**

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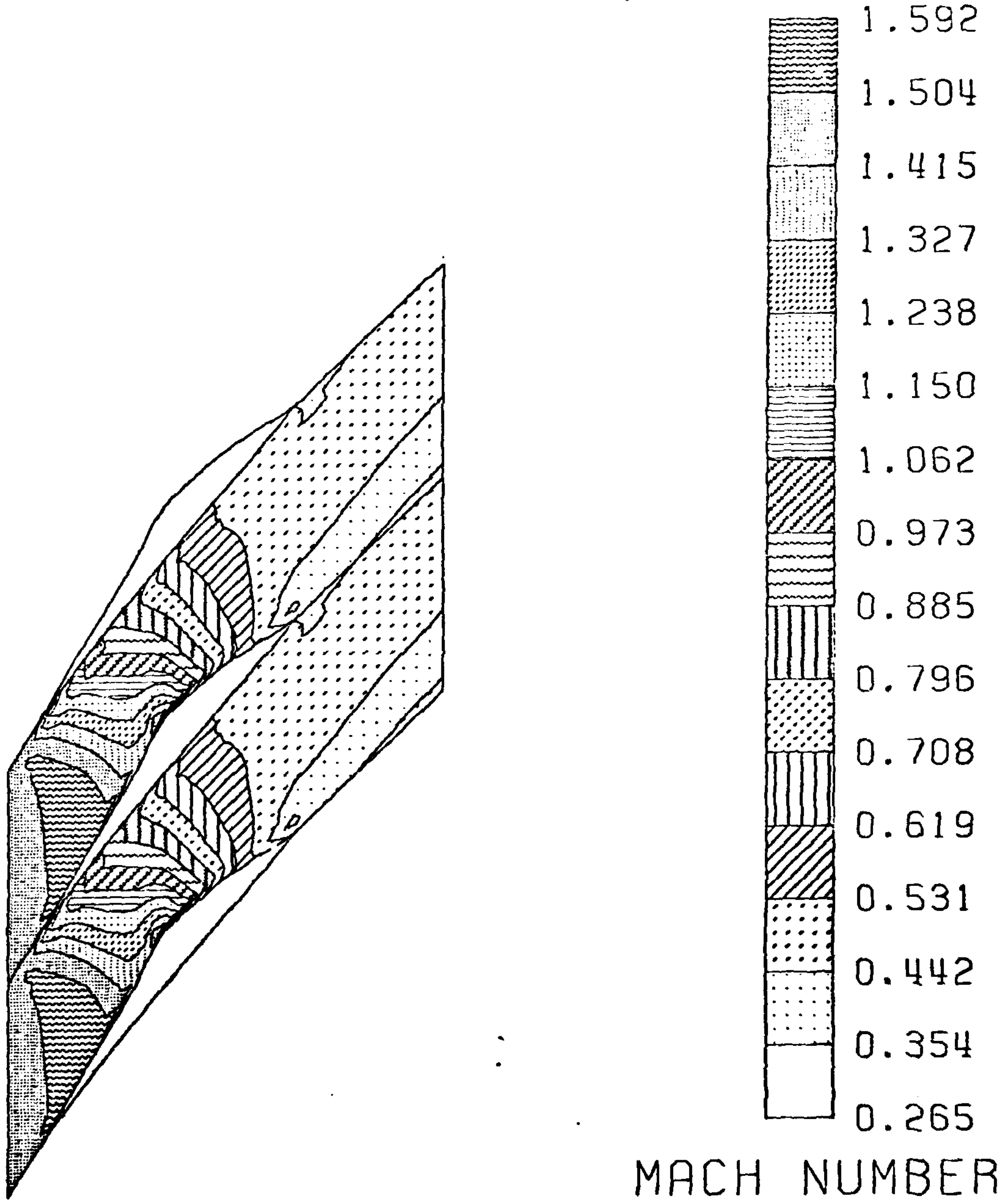


Figure 1.2 PRE-COMPRESSION BLADE

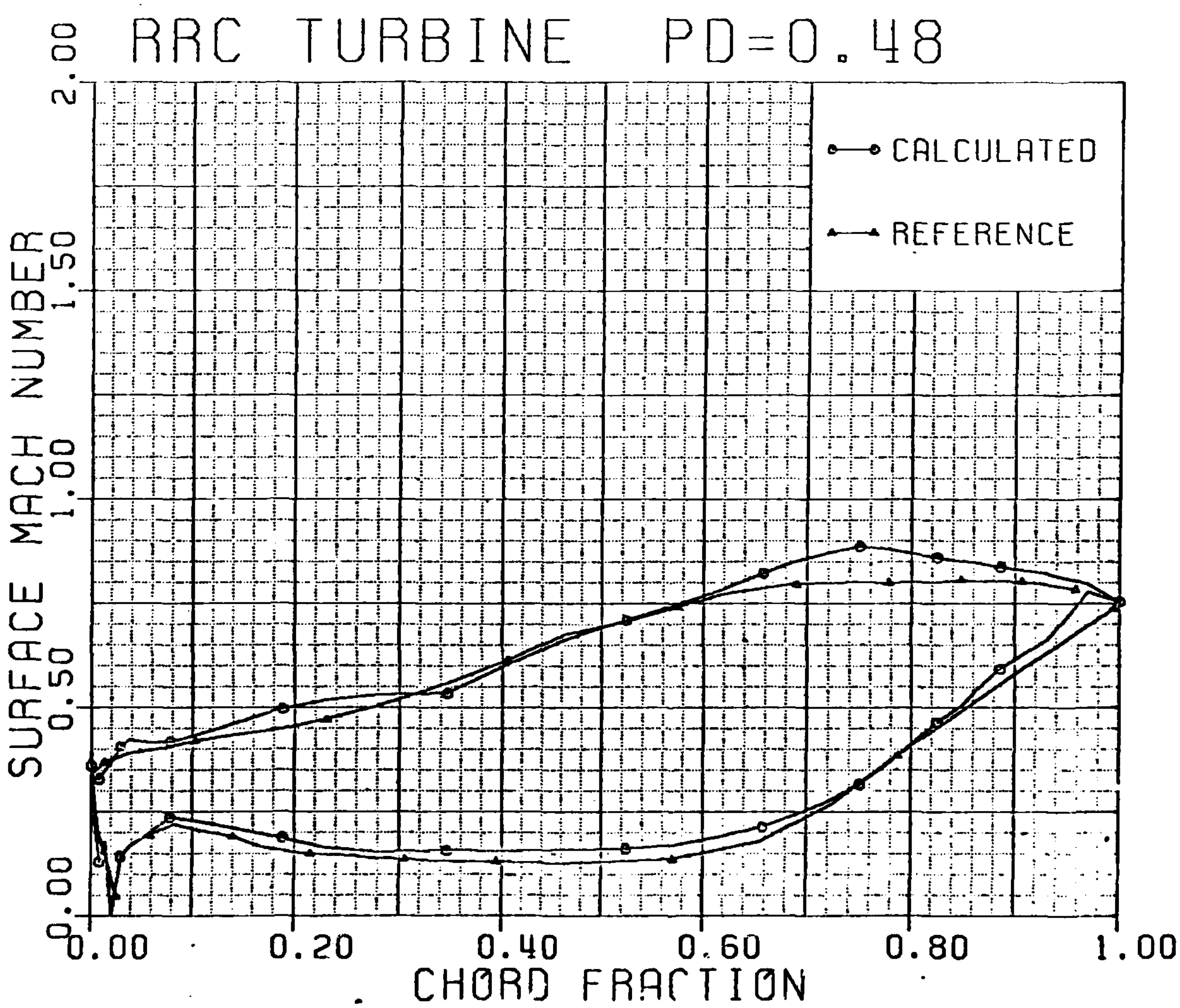


Figure 1.3 Comparsion of experimental data and calculated result.

B. Advanced Measurements in Transonic Compressors

The research effort is aimed at elucidating design point operation loss mechanisms of high work, high mass flow transonic compressor stages. In particular, the Air Force Aero Propulsion Laboratory High [illegible] ~~Three~~ Flow (HTF) stage is being examined in the M.I.T. Blowdown Compressor Facility.

Work in this reporting period has continued on the data reduction of the M.I.T. time resolved measurements of the rotor outflow and comparisons with steady state measurements taken at AFAPL. An example is shown in Figure 1.4 which is a comparison of the spanwise pressure distribution of the time average of the time resolved M.I.T. data and the conventional pitot total measurements from AFAPL. Note that these measurements are generally in agreement except at mid-span where the time resolved pressures are low. This discrepancy has been seen in a similiar comparison at M.I.T. with a different compressor stage from NASA Lewis Research Center. It is thought to result from the termination of the 3-D bow shock as the relative blade Mach decreases from tip to hub.

This low pressure region is also a low efficiency region when the total temperature is calculated from the data using Euler's Turbine Equation. These high losses do not appear in the conventional measurements. The Euler equation assumes steady flow (as do the conventional measurements), however the time resolved measurements show the flow to be unsteady. Thus, there is a requirement to directly measure time resolved total temperature.

The second effort under this task is the development of a new technique to measure fluctuating total temperature. The full scale dual wire asperating probe (Figure 1.5) has been constructed and tested in a free air jet, with various electrical mechanical problems having been overcome. It will be tested in the the compressor in early June.

In order to decrease the change in corrected speed of the compressor during the test time, a tungsten flywheel will be added to the facility in July (pending AFOSR approval). During this time, the dual wire asperating probe will be recalibrated in an argon-freon jet (the tunnel test gas) using the blowdown tunnel as a test reservoir.

After the flywheel is unstalled, the high response data will be remeasured. The temperature measurements will then be taken behind the rotor stator.

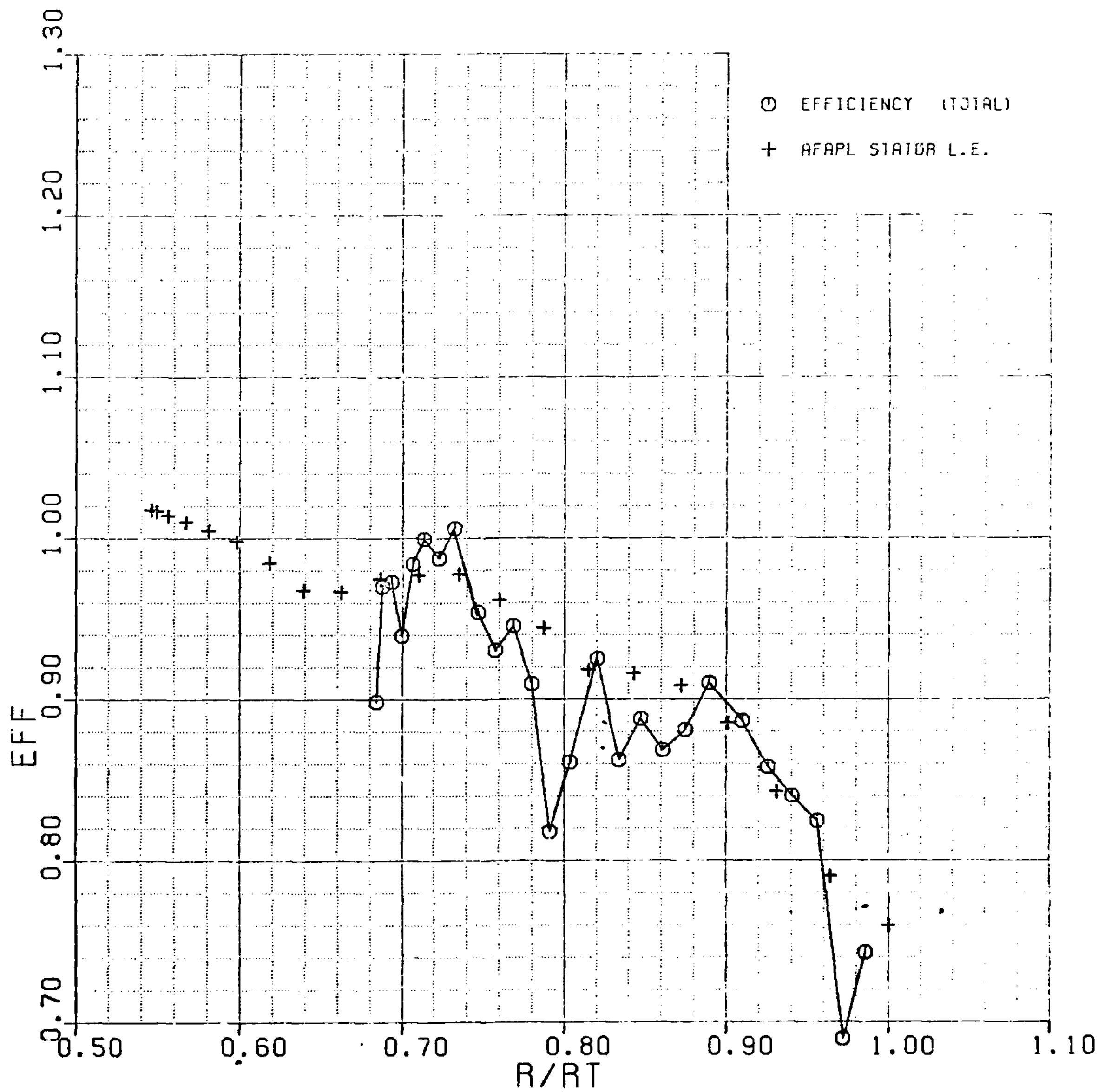


Figure 1.4 Efficiency versus Radius for AFAPL HTF Stage

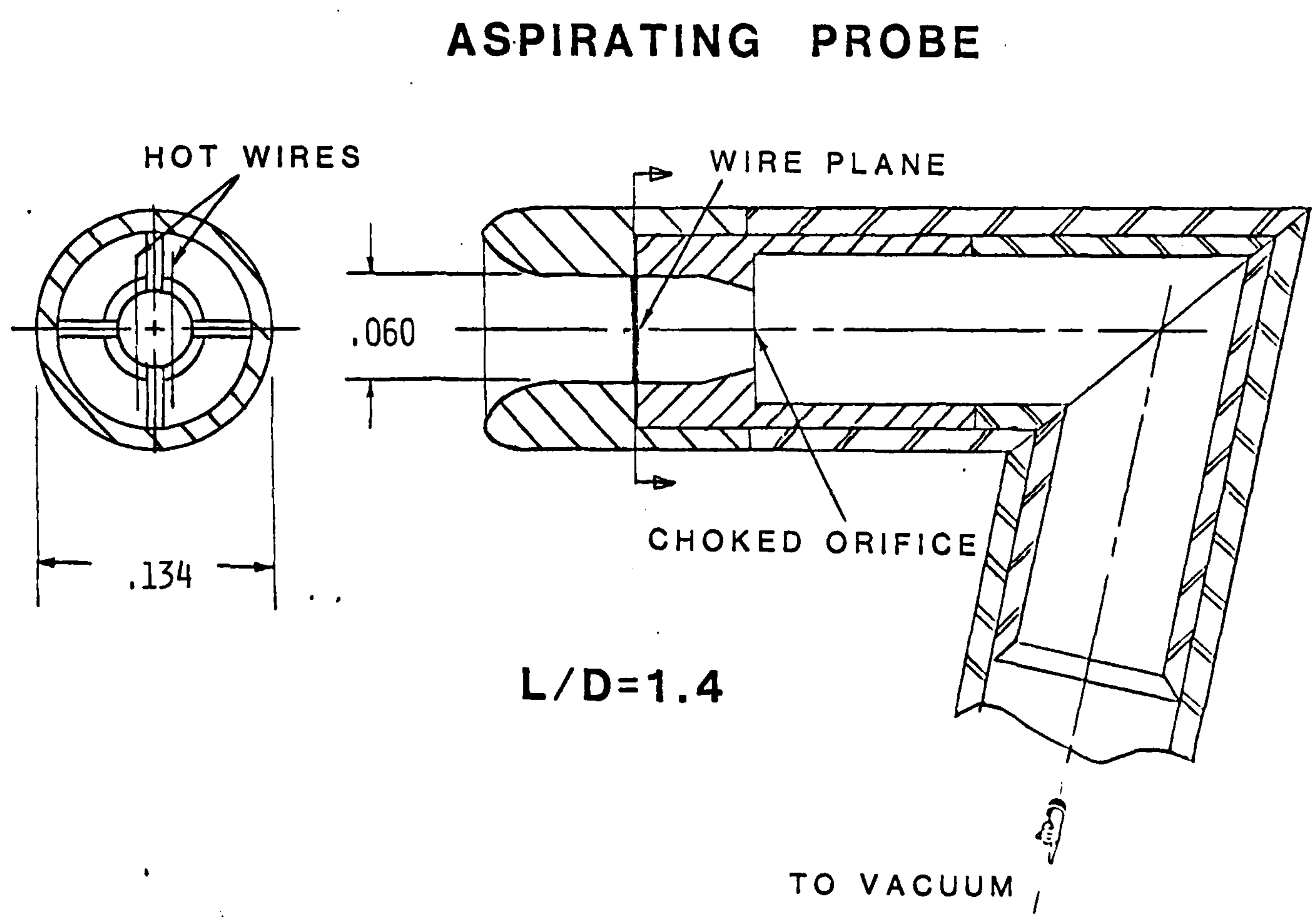


Figure 1.5 Aspirating Probe

Task II: Mechanisms of Compressor Stability Enhancement Using Casing/Hub Treatment

During the past six months we have carried out the studies for the design of a wall stall configuration. Several concepts have been incorporated into this. First, the stagger of both rotor and stator have been increased from that tested by Prell [2.1]. (This has been made possible through the use of the exit fan, which we obtained under AFOSR sponsorship.) This has the effect of reducing the incidence for a given value of $\Delta P/q$ - i.e. the same turning at lower stagger produces a higher $\Delta P/q$. This also tends to help in terms of lowering the ratio of D-factor to $\Delta P/g$.

In addition to this, the tip clearance has been increased by 0.030". This should have the effect of decreasing the peak pressure rise that can be achieved at stall with an endwall stall configuration. As shown by Koch[2.2] (for multistage compressors) this effect can be substantial.

The increased rotor stagger necessitated a new set of inlet guide vanes. For simplicity these have been made of sheet metal. They are of guide high solidity, so that no large amount of separation is expected in spite of the high turning. This high solidity will also act to decrease the IGV wake/rotor noise which has been unacceptable. A typical calculation of the D-factor and $\Delta P/q$ profiles are shown in Figures 2.1 and 2.2. In particular note that the value of $\Delta P/q$ at the 90% immersion location is 0.48 compared to the peak realizable pressure rise of 0.39 that is predicted by the Koch correlation, i.e., we are above the wall stall boundary based on the above conditions. It should be noted however that due to the strong skew (distortion) in the stator inlet total pressure profile the $\Delta P/q$ at midspan is considerably lower (0.29). The incidence angles on the stator are on the order of a degree over most of the blade, considerably less than those found in the configuration used by Prell, and the calculated D-factor at 90% immersion is 0.65 compared to the calculated value of 0.74 in the Prell compressor. From these indications therefore it seems as if the use of the increased stagger has pushed us much further toward a wall stall situation.

One further part of the work done on this project is the construction of an acoustic enclosure for the inlet of the compressor. This is a 2.5 meter high octagon which is lined with acoustic treatment on the inside. This is now almost completed; however we plan to run without it if it is not ready.

The situation now is that the stage has been built up and we are preparing to take data using the smooth hub and the hub casing treatment. As far as the theoretical aspect of this work, we are exploring use of a simple control volume approach to the casing boundary layer growth through the rotor. Preliminary results indicated that fluid in the momentum transfer to the endwall region (due to the fluid passing in and out of the grooves) can have a very substantial effect on the blade row exit boundary layer thickness.

References

2.1. Prell, M.E., 'An Experimental Study of Stator Hub Casing Treatment", M.I.T. GT&PDL Report No. 161, July 1981.

2.2. Koch, C.C., "Stalling Pressure Rise Capability of Axial Flow Compressor Stages", ASME J. Eng. Power, Vol. 103, pp. 645-656, October 1981.

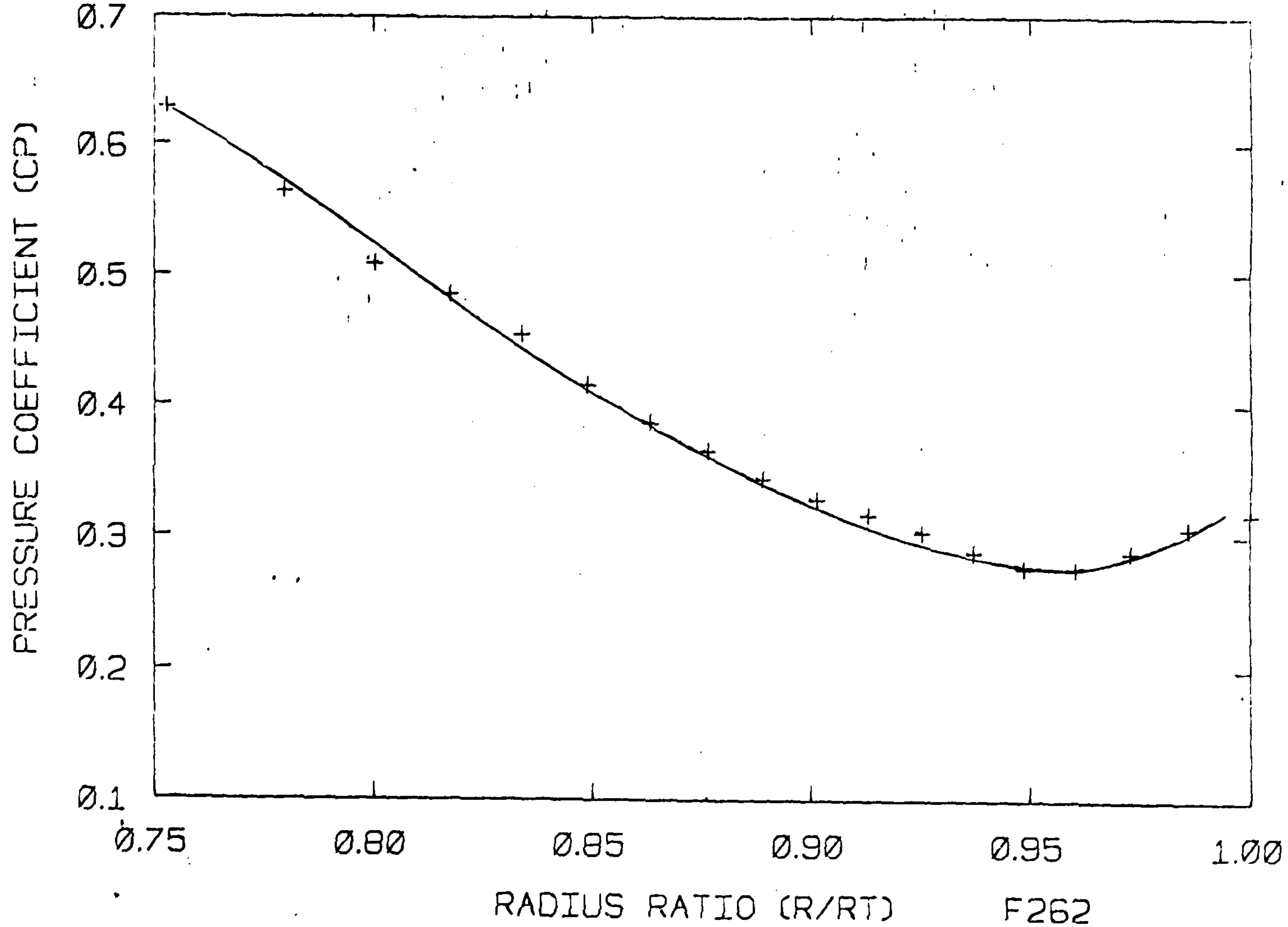


Figure 2.1 Stator ΔP/q versus radius ratio.

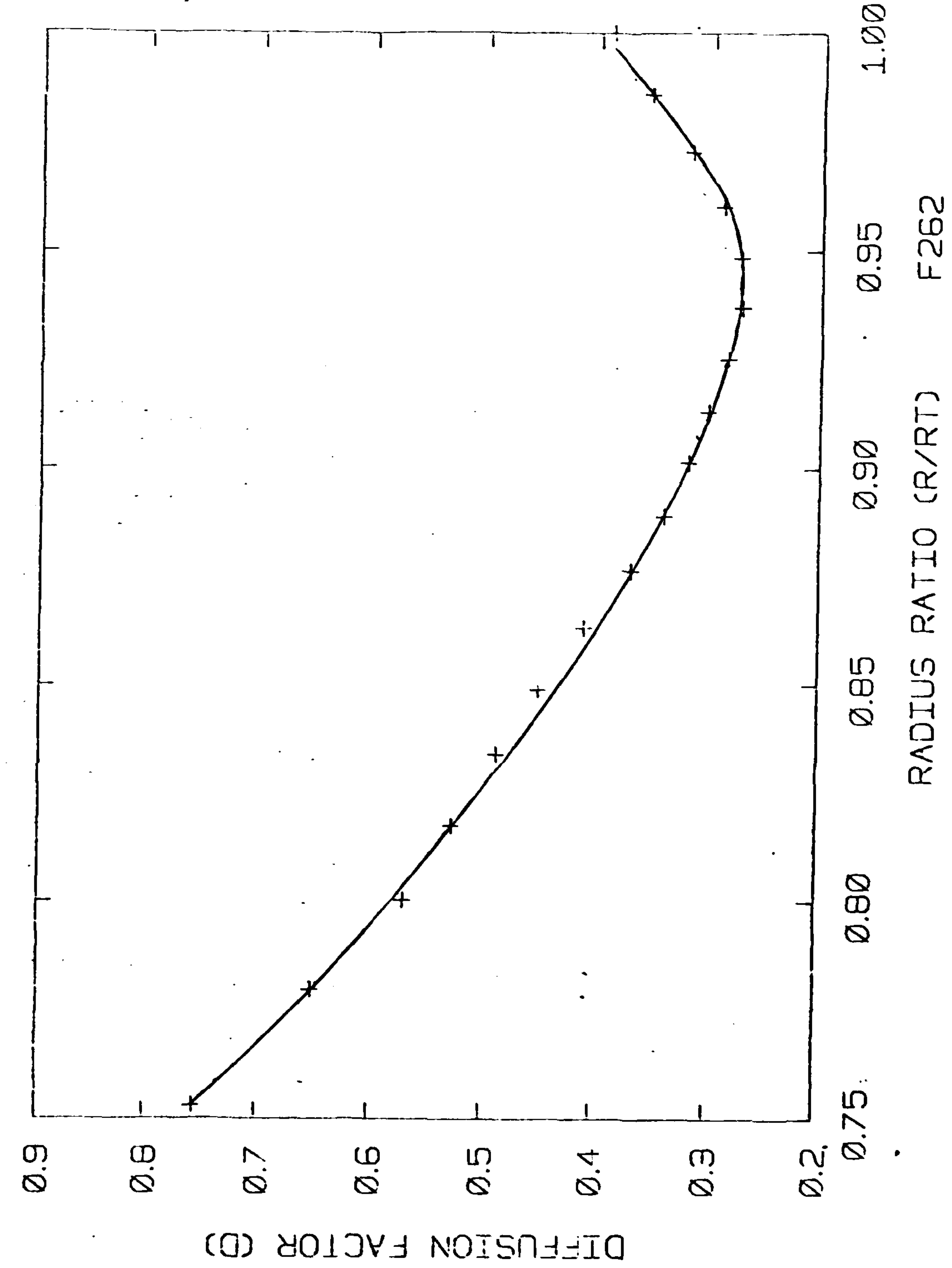


Figure 2.2 Stator D-factor versus radius ratio.

Task III: Inlet Vortex Flow Distortion

During the past six months we have started to take aerodynamic data using our .15 m diameter inlet in the Wright Brothers Wind Tunnel. This inlet had been designed using an axisymmetric potential flow code (3.1) and the experimentally determined boundary layer separation limits described in (3.2). We have actually fabricated two of these inlets. One, made of plexiglass, is to be used for flow visualization. The other, made of aluminum, has forty surface pressure taps in order to measure the static pressure distribution round the inlet. The initial experiments involved tunnel flow field mapping and flow visualization (smoke and surface streamlines) to ensure that we indeed had a vortex.

Typical results from the tunnel flow field mapping are presented in Figure 3.1 which shows the velocity profile across the tunnel as a function of height at one axial station. The test section velocity is roughly 6 m/s.

During this time period we have also designed and fabricated a "grid" of slats of varying area for the generation of a specified shear profile. A photograph of this grid installed in the tunnel is shown in Figure 3.2. The total pressure distribution that is produced by this grid is plotted in Figure 3.3. In the figure the horizontal axis is distance across the tunnel and the vertical axis is the non-dimensional (local) dynamic pressure, normalized by the dynamic pressure at the middle of the tunnel. The measurements were made at an axial location corresponding to the inlet lip, 1.22 m (12 slat spacings) downstream of the grid. The different symbols correspond to different heights above the ground. Note that these have been chosen to be either behind the slats or between them. It can be seen that there is no systematic difference between the various levels, with the exception of the lowest, which is in the ground boundary layer, and over the center four feet of the tunnel the profile is roughly linear. Since, the capture area is less than this, the profile is considered adequate, at least for the present series of tests.

One result of the flow visualization studies is shown in Figure 3.4. This is a photograph of surface ink flow on a mylar sheet which was placed on the ground plane under the inlet. The position of the inlet and the ambient flow direction are marked. The spiral pattern associated with the vortex can be seen clearly.

The first set of experiments were carried out to examine the flow regimes that were present. These revealed that at some conditions the inlet was in the transition regime (between subcritical and supercritical cylinder flow) and the pressure distributions were thus quite sensitive to small surface irregularities. To overcome this we are currently investigating the use of different boundary layer "trips". Once this is completed we will focus on acquiring quantitative data on the static pressure distribution round the inlet as well as on the details of the downstream trailing vortex flow field. It is antipated that this will form the major part of our effort over the next several months.

References

3.1 Stockman, N.O. and Farrell, C.A., "Improved Computer Programs for Calculating Potential Flow in Propulsion System Inlets", NASA TM-73728, July 1977.

3.2 Boles, M.A., and Stockman, N.O., "Use of Experimental Separation Limits in the Theoretical Design of V/STOL Inlets", AIAA Paper 77-878, presented at Thirteenth Propulsion Conference, July 1977.

TUNNEL TEST SECTION TOTAL PRESSURE DISTRIBUTION 1.5 M DOWNSTREAM FROM LEADING EDGE

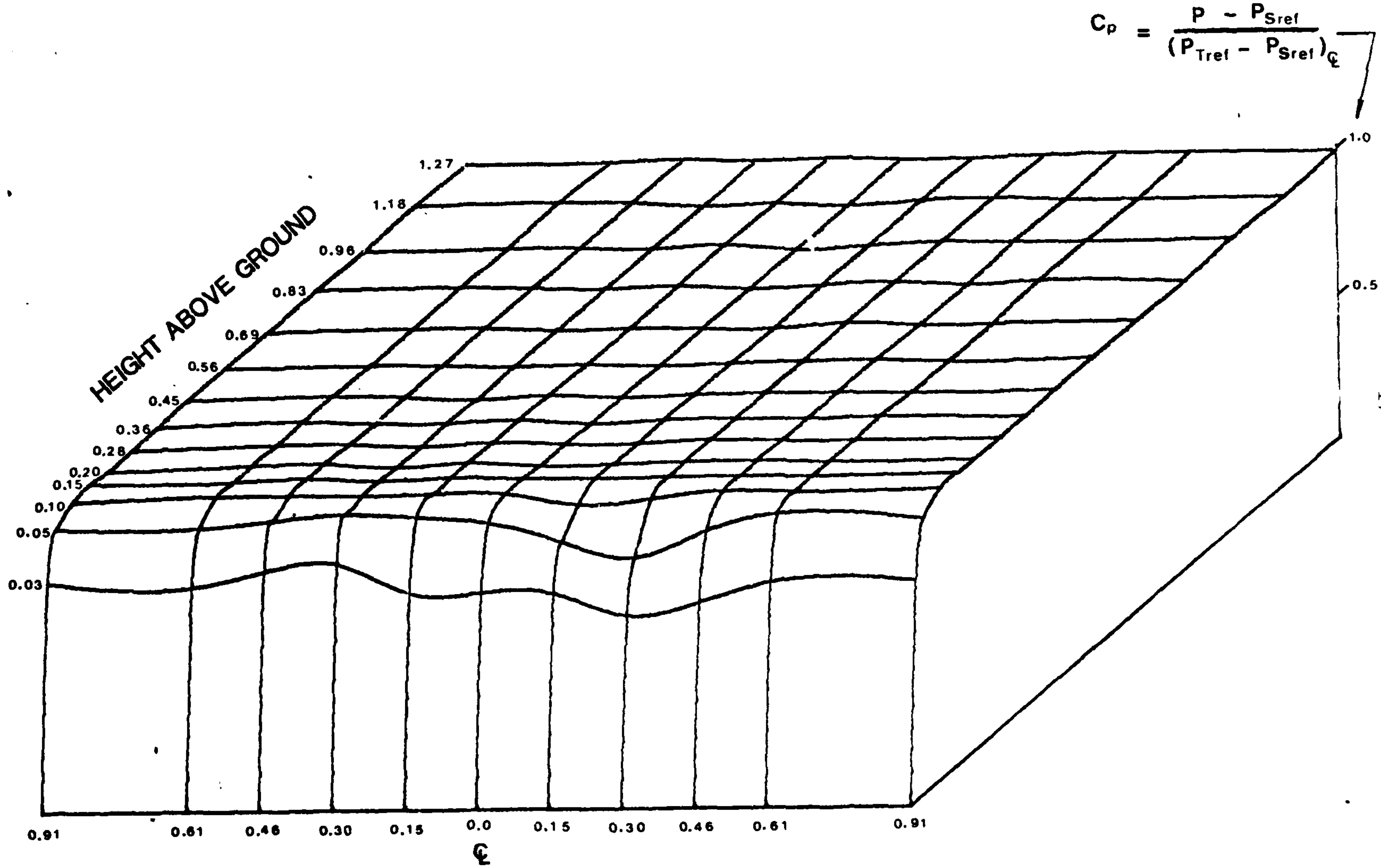
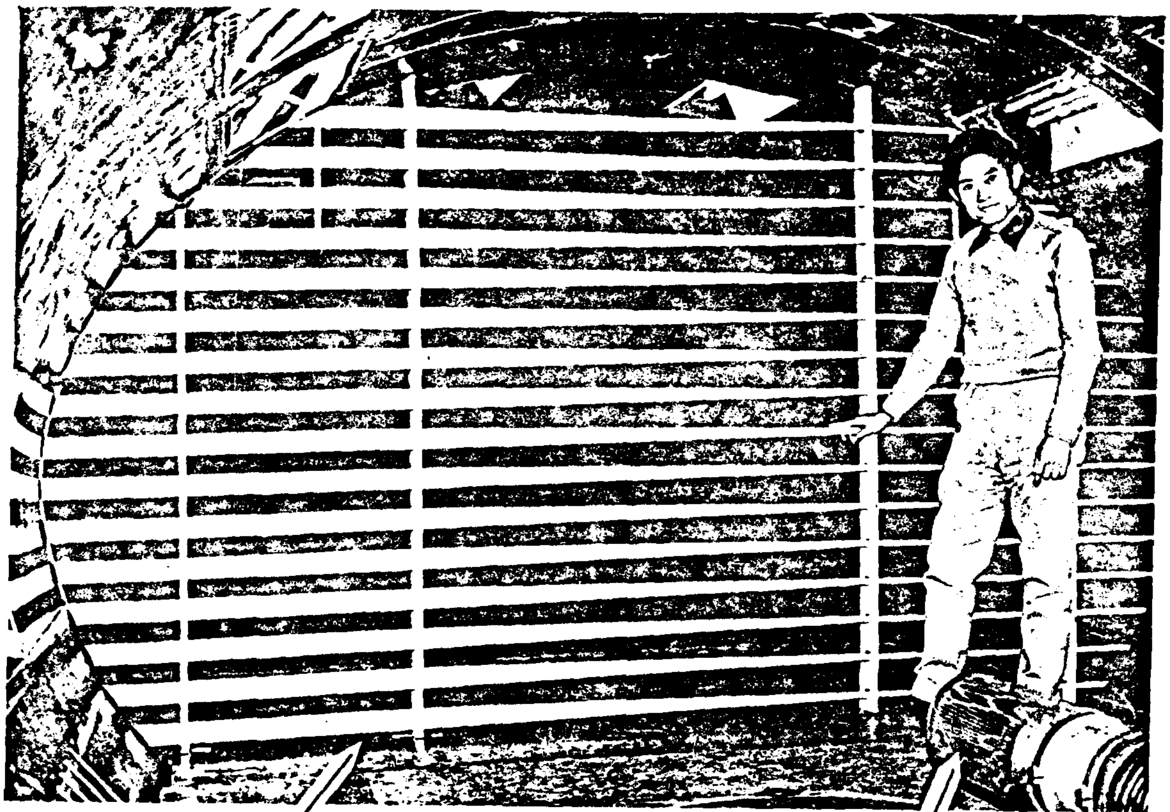


Figure 3.1 DISTANCE ACROSS TUNNEL



Grid of Wooden Spires
for Shear Profile Generation

Plexiglas Inlet

Fig 3.2 WIND TUNNEL SHEAR PROFILE GENERATOR

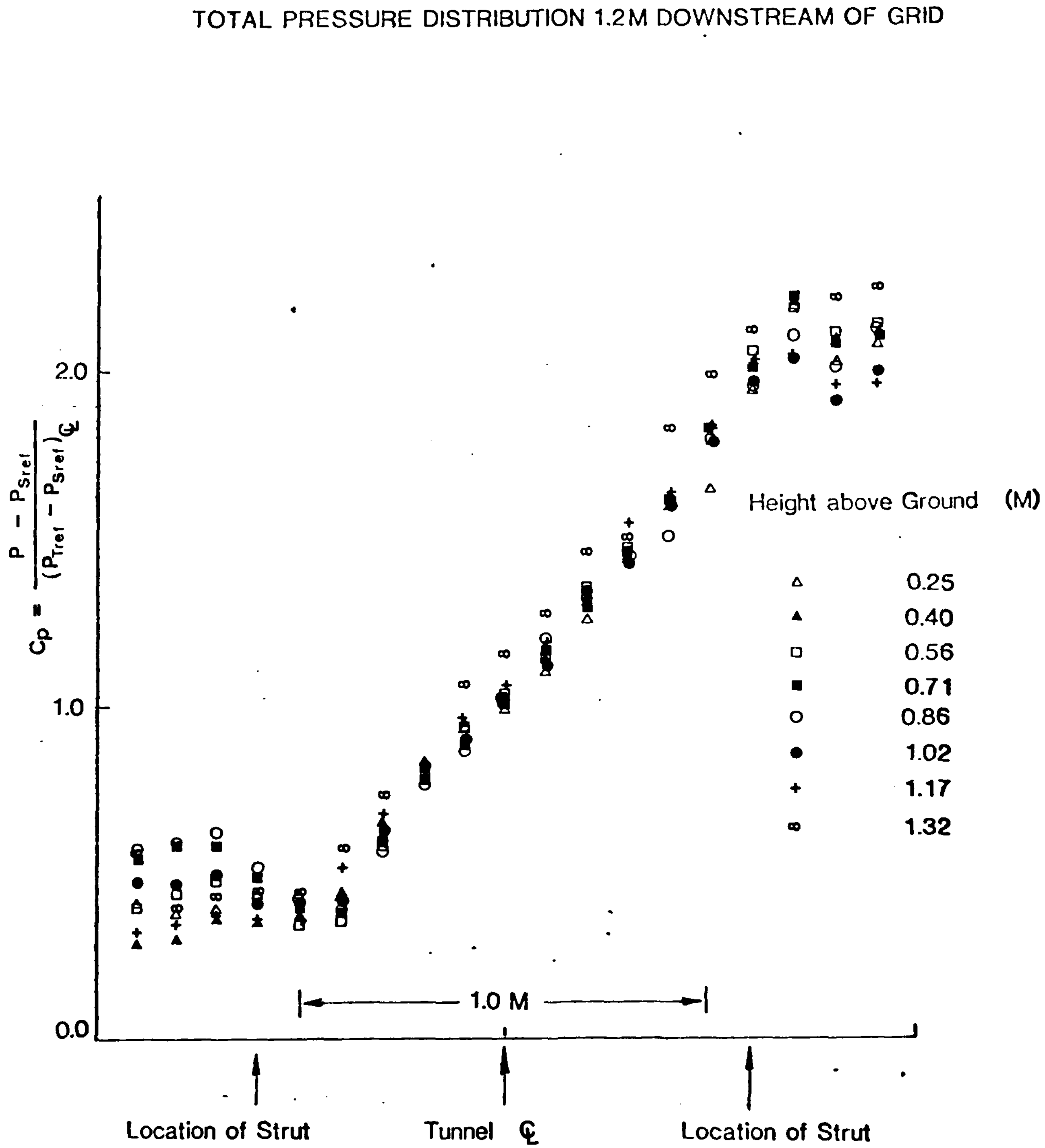


Figure 3.3 DISTANCE ACROSS TUNNEL

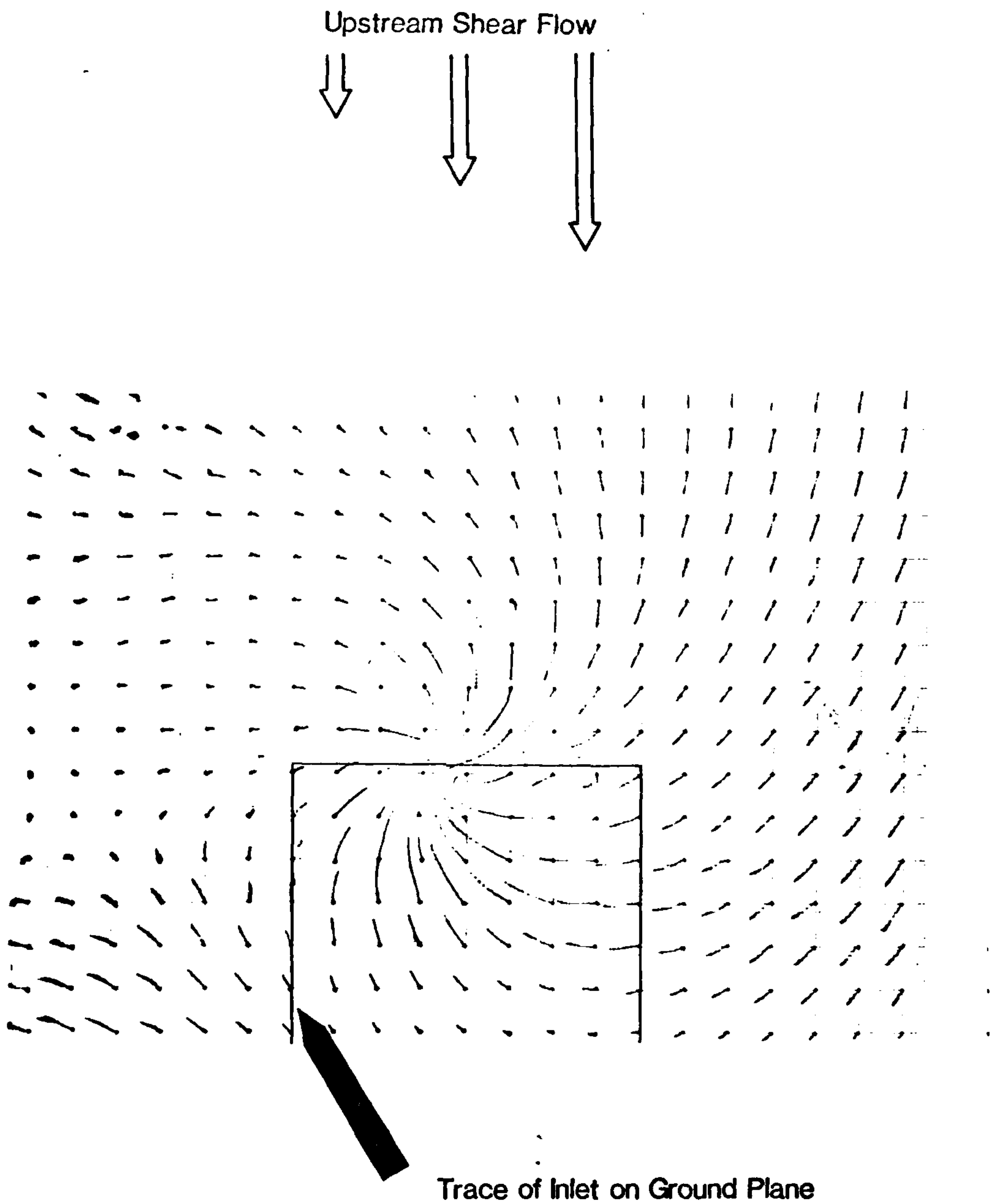


Fig 3.4 SKIN FRICTION LINES SHOWING GROUND VORTEX

Task IV: Investigations of 3-D Flows in Highly-loaded Turbomachines

A: Blade-to-Blade Flow Analysis

The following paragraphs describe progress made since October 1981 in studying three-dimensional, (especially blade-to-blade) effects in practical turbomachine flows of high loading. Part of the emphasis is on contributing to improved design techniques.

Successful calculation of incompressible, non-linear three-dimensional flow associated with high-swirl, "very thin" blades in cylindrical annuli was reported previously. Those results confirm the expectation of important implications of 3-D flow on blade design.

Over the past several months the main effort has been toward expanding the types of situations that can be treated viably. Using a new analytical technique (which we refer to as "smoothing"), we have studied, in varying degrees of completeness the following areas.

- Thickness effects on blade camber -- or, for given blade shapes, on loading
- Blade Design in Compressible Flow
- Variable Wall (Hub-Shroud) Geometry

We have also verified that "smoothing" reproduces results already know for appropriate special cases.

1. Thickness

The choice of blade profile is known to affect the amount and form of blade camber needed to accomplish a given amount of turning of the flow in a turbomachine.

An extensive study (so far limited to the 2-D cascade example to test the method) of swirling blade-to-blade flow as effected by blade thickness has been undertaken. Thickness (or "blockage") effects have been found to significantly modify blade-to-blade results for realistic loadings.

The main emphasis here has been to learn how to produce a readily-useable tool to provide rational blade design (choice of camber) including actual profile shapes. Extension to 3-D is expected to follow readily.

2. Compressible Flow

The "smoothing" procedure may be of use in simplifying design predictions in compressible flow. Combining the technique with the Clebsch transformation appropriate for complicated turbomachine geometries can help to identify which compressibility effects predominate in affecting blade design choices. This suggests that a simplified (but still definitely "non-linear") picture of 3-D compressible flow past turbomachine blades can be constructed. This study is still in its early stages.

3. Variable Wall Geometry

Hub and shroud shapes in turbomachines vary considerably from pure cylindrical even in axial turbomachines.

A method has been devised to construct a "pseudo-flow" in essentially arbitrary duct geometry, the "streamlines" and "potential lines" of which provide a convenient orthogonal network for the useful description of the real flows of interest in the same geometry.

This transformation technique appears to have different degrees of usefulness, depending on whether it is to be applied to axial or radial machines.

B: Numerical Study of Secondary Flow in a Bend Using Spectral Methods

During the past 6 months, we have formulated the numerical algorithm for solution of the incompressible Navier-Stokes equations for flow in a bend of square cross-section. Specifically, we have chosen to write the incompressible Navier-Stokes equation in rotational form as follows:

$$\frac{\partial \vec{v}}{\partial t} = \vec{v} \times \vec{w} - \nabla\pi + \nu\nabla^2\vec{v} \tag{1}$$

$$\nabla \cdot \vec{v} = v \tag{2}$$

where $\vec{v}$ is the velocity $\vec{w}$ the vorticity, π the stagnation pressure normalized by the density, and ν the kinematic viscosity. A cylindrical co-ordinate system (r,θ,z) is used with the square section of the bend lying on the r-z plane as shown in Fig. 1.

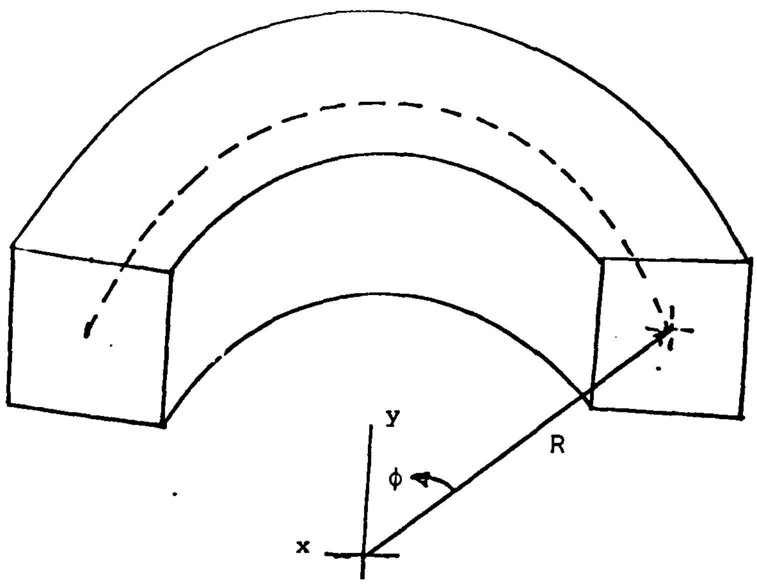


Figure 1

The approach followed is to expand the velocity field $\vec{v}$ and the stagnation pressure field π in triple Chebyshev series as follows:

$$\vec{v}_{ijk} = \sum_{\ell=0}^{L} \sum_{m=0}^{M} \sum_{n=0}^{N} \vec{u}_{\ell mn}(t)\, T_\ell(R_i)\, T_m(\phi_j)\, T_n(Z_n) \qquad (3)$$

$$\pi_{ijk} = \sum_{\ell=0}^{L} \sum_{m=0}^{M} \sum_{n=0}^{N} P_{\ell mn}(t)\, T_\ell(R_i)\, T_m(\phi_j)\, T_n(Z_n) \qquad (4)$$

$T_\ell(x)$ denotes a Chebyshev polynomical of argument x and degree ℓ. In expressions (3) and (4), we have mapped the (r,θ,z) domain appropriately into (R,ϕ,Z) domain so that the computational domain lies within $|R| \le 1$, $|\phi| \le 1$ and $|z| \le 1$. The mapping functions used are

$$R = \left(\frac{2}{r_o - r_i}\right) r - \left(\frac{r_o + r_i}{r_o - r_i}\right) \qquad (5)$$

$$\phi = 2\,\frac{\theta}{\theta_o} - 1 \qquad (6)$$

where r_i and r_o are the inner radius and outer radius of the bend while θ_o is the total angle of the bend. By choosing the collocation points (i,j,k) at

$$R_i = \cos\frac{\pi i}{L} \qquad (7a)$$

$$\phi_j = \cos\frac{\pi j}{M} \qquad (7b)$$

$$Z_k = \cos\frac{\pi k}{N} \qquad (7c)$$

we can efficiently evaluate (3) and (4), both in the Chebyshev and physical space, using FFT techniques. In addition, we achieve high grid resolution at the wall boundaries which is essential for good viscous layer resolution.

The fractional-time stepping technique is employed to advance the Navier-Stokes equation in time as follows:

(1) The convective (or advective) part:

The first fractional step is the convective part. A semi-implicit Crank-Nicholson-Adams-Bashforth Scheme is used and yields

$$\frac{\vec{v}^{(1)n+1} - \vec{v}^{\,n}}{\Delta t} + \frac{1}{2}\frac{\bar{v}_\theta}{r}\frac{\partial \vec{v}}{\partial \theta}^{(1)n+1} = \frac{3}{2}(\vec{v} \times \vec{w})^n - \frac{1}{2}(\vec{v} \times \vec{w})^{n-1} \tag{8}$$

Implicit Crank-Nicholson for stabilization of time stepping

$$+ \frac{\bar{v}_\theta}{r}\frac{\partial \vec{v}^{(1)n}}{\partial \theta} - \frac{1}{2}\frac{\bar{v}_\theta}{r}\frac{\partial \vec{v}^{(1)n-1}}{\partial \theta}$$

Explicit Adams-Bashforth part

$\bar{v}_\theta$ is the value of the bend through-flow velocity at the inflow boundary. Superscript (1) denotes the first fractional time step while superscript n denotes the time level. For this part, no outflow B.C. are required as the spectral method does not introduce spurious B.C. The inflow boundary conditions are:

$$\begin{aligned} v_r &= 0 \\ v_\theta &= C_o(1 - R^2)(1 - Z^2) \\ v_z &= 0 \end{aligned} \tag{9}$$

Because $\bar{v}_\theta = \bar{v}_\theta(R,Z)$, equation (8) yields spectral-space equations for the chebyshev coefficients in the ϕ-direction. These equations forme an almost tridiagonal system and hence can be solved efficiently.

(2) The pressure correction step:

The second fractional step enforces the incompressibility condition and the normal wall boundary conditions, which are not required by the first fractional step. This is achieved via the solution of π from

$$L\,[\pi] = \frac{1}{\Delta t}\ \nabla \cdot \vec{v}^{\,(1)n+1} \tag{10}$$

where L is the mapped Laplacian operator in (R,ϕ,Z) space. The boundary conditions imposed on π are:

$$\frac{\partial \pi}{\partial R} = \frac{(r_o - r_i)}{2\Delta t}\, v_r^{(1)n+1} \qquad \text{at} \quad R = \pm 1 \tag{11a}$$

$$\frac{\partial \pi}{\partial z} = \frac{1}{\Delta t}\, v_z^{(1)n+1} \qquad \text{at } Z = \pm 1 \tag{11b}$$

$$\frac{\partial \pi}{\partial \phi} = \frac{1}{2}\,\frac{\theta_o}{\Delta t}\left(\frac{v_o - v_i}{2} R + \frac{r_o + r_i}{2} \right) v_\theta^{(1)n+1} \qquad \text{at inflow boundary} \tag{11c}$$

$$\frac{\partial \pi}{\partial \phi} = 0 \qquad \text{at the outflow boundary} \tag{11d}$$

(Note that the imposed outflow boundary condition, Equation (11d), is not strictly correct, and we will be attempting to improve (11d) utilizing a linearized secondary flow analysis.) In eqn. (11), we have used subscript (r,θ,z) to denote vector component in the r-,θ-,z-direction. The corrected velocity field $\vec{v}^{\,(2)n+1}$ is obtained through

$$\vec{v}^{\,(2)n+1} = \vec{v}^{\,(1)n+1} - \Delta t \nabla \pi \tag{12}$$

The tensor-product technique and the spectral iteration technique is used to solve for π from the Poisson equation (10). In this combined method, we used matrix diagonalization in the ϕ- and z- direction and spectral iteration in the radial direction. *This method required a preprocessing* step done at the very *beginning.*

(3) The viscous correction step:

The final fractional step consists of imposing the no-slip conditions at the walls together with appropriate inflow and outflow B.C. The backward Euler-like scheme is used and it yields

$$\frac{v_r^{n+1} - v_r^{(2)n+1}}{\Delta t} - \nu\left(\nabla^2 v_r^{n+1} - \frac{v_r^{n+1}}{r^2} \right) = -\frac{2\nu}{r^2}\frac{\partial v_\theta^{(2)n+1}}{\partial\theta} \qquad (13)$$

$$\frac{v_\theta^{n+1} - v_\theta^{(2)n+1}}{\Delta t} - \nu\left(\nabla^2 v_\theta^{n+1} - \frac{v_\theta^{n+1}}{r^2} \right) = \frac{2\nu}{r^2}\frac{\partial v_r^{n+1}}{\partial\theta} \qquad (14)$$

$$\frac{v_z^{n+1} - v_z^{(2)n+1}}{\Delta t} - \nu\nabla^2 v_z^{n+1} = 0 \qquad (15)$$

This step involves the solution of 3 poisson like equations. Again, as in the solution for π, we used a combination of the tensor-product method and the spectral iteration method (in the radial direction) for the solution of v_r^{n+1}, v_θ^{n+1} and v_z^{n+1}.

Steps (1) and (2) are of $O(\Delta t^2)$ while step (3) is of $O(\nu\Delta t)$. However, due to the curvature term, the *pressure and* the *viscous* operator do not commute, therefore the overall accuracy will be of $O(\Delta t)$. However, $O(\Delta t^2)$ maybe achievable by alternating the direction of fractional stepping. It should also be mentioned that the implementation of fractional step (1) and (2) only, will yield an inviscid solution of the flow.

Current status is that a large part of the software has been written including programs: (1) to compute and invert triple Chebyshev series efficiently using existing FFT software, (2) to solve three-dimensional Poisson equation in the cylindrical coordinate system with both Neumann type of B.C. (pressure step) and Dirichlet B.C. (viscous step) using tensor-product technique and spectral iteration technique and (3) to implement the above three fractional steps. Currently, we are testing these programs against model equations and functions with known analytical answers to ensure the correctness of the written software as well as to find the eigensolutions of the spectral iteration operator with Neuman type of B.C. Once this phase is over, we are then in a position to synthesize the various parts of the program to time step the Navier Stokes Equation for flow in a bend.

GENERAL PROGRESS ON AFRAPT

During the last year the Air Force OSR sponsored educational program, called Air Force Research in Aeronautical Propulsion Technology (AFRAPT), was initiated. An industry-faculty steering committee was established. It consists of

Dr. Michael Salkind	AFOSR, Chairman
Mr. Walter Schrader	AVCO Lycoming
Mr. Lynn Snyder	DDA
Mr. Elmer Wheeler	Garrett Aerospace Corp.
Mr. Fred Erich	GE Lynn
Mr. David Wisler	GE Evandale
Mr. Morris Zipkin	Pratt and Whitney (GPD)
Professor Sandford Fleeter,	Purdue University
Professor Peter Jenkins,	Texas A & M University

and

Professor E.E. Covert,	Massachusetts Institute of Technology, Secretary

This group has met twice, established procedures, put out the first year's brochure and generally put the program in motion.

This spring 22 applications were received, 9 offers were made and at the present time 6 acceptances have been received.

The Steering Committee will meet shortly to review the program and plan for next year.

3. JOURNAL PUBLICATIONS

1. Thompkins, W.T. and Tong, S.S., "Inverse or Design Calculations for Nonpotential Flow in Turbomachinery Blade Passages", ASME J. Eng. for Power, Vol. 104, pp. 281-286, April 1982.

2. Thompkins, W.T., and Tong, S.S., "A Design Calculation Procedure for Shock-Free or Strong Passage Shock Turbomachinery Cascades", ASME paper 82-GT-220.

3. De Servi, F., Vigiuer, H.C., Greitzer, E.M., and Tan, C.S., "Mechanisms of Inlet Vortex Formation", submitted to J. Fluid Mechanics.

4. Tan, C.S., Hawthorne, W.R., and McCune, J.E., "Three-Dimensional Blade Design using an Analytic Theory", to be submitted for publication to ASME 1983 Gas Turbine Conference.

4. PROGRAM PERSONNEL

Principal Investigators:

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Senior Lecturer

Graduate Students:

9/80 - Present	Peter Cheng
9/81 - Present	Thong Dang
9/81 - Present	Philippe Kletzkine
9/80 - Present	Wen Liu
9/80 - Present	Wing-Fai Ng
9/79 - Present	Siu Shing Tong

5. INTERACTIONS

Presentations

E.M. Greitzer, "The Stability of Pumping Systems," seminar presented at Texas A & M University, December 1981.

E.M. Greitzer, lecture series on "Flow Instabilities in Gas Turbine Engines," (including inlet distortion effects, stall enhancement using casing treatment, non-uniform swirling flows in turbomachines, etc.), presented at Nanjing Aeronautical Institute, Nanjing, China, and Beijing Institute of Aeronautics and Astronautics, Beijing, China, October, 1981.

E.M. Greitzer, seminar at NASA Lewis Research Center "Non-Recoverable Stall in Axial Compressors," January, 1982.

A.H. Epstein, "Time Reserved Measurements in a Transonic Compressor," paper presented at 1982 ASME Gas Turbine Conference, London, England, April, 1982.

As described in previous reports there are strong interactions between the Gas Turbine Laboratory and the aircraft engine industry, as well as Air Force and other government laboratories. In addition we have an active seminar program to bring speakers from industry and/or government to M.I.T. During the past six months these have included

Mr. B.A. Robideau, Pratt & Whitney Aircraft, Commercial Products Division, "Practical Application of Aerodynamic Principles to Multistage Compressor Design"

Dr. R. Mani, Corporate Research Center, General Electric Company, "Some Aspects of Gas Turbine Engine Post-Stall Behavior"

Dr. M. Sajben, McDonnell-Douglas Corporation, "Unsteady Aspects of Internal Flows with Shock/Boundary Layer Interaction"

Dr. R. Norton, Rolls Royce U.S.A. (on leave at M.I.T. Department of Aero/Astro), "Transonic Flow in a Nozzle Guide Vane"

Mr. R. Novak, Aircraft Engine Group
General Electric Company
"A Mixed-Flow Cascade Passage Design Procedure Based on a Power Series Expansion"

Dr. J. Caspar, United Technologies Research Center
"Numerical Calculation of Cascade Flows Using Non-Orthogonal Grids"

6. DISCOVERIES, INVENTIONS AND SCIENTIFIC APPLICATIONS

During the work period covered by this report, there have been no new inventions or discoveries.

7. CONCLUSIONS

In brief, although some of the projects appear farther along than others the M.I.T. Multi-Investigator effort does appear to be basically on schedule. In addition if one looks at the experimental part of this effort not only for this contract period but also the last, it appears that the emphasis has shifted from (the initial) building up of rigs and obtaining data requisition equipment to the analysis and interpretation of the aerodynamic data.

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